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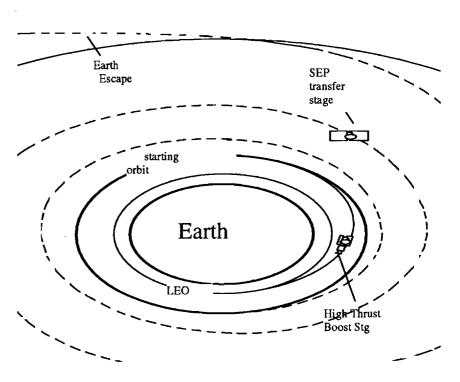
HUMAN MARS TRANSPORTATION APPLICATIONS USING SOLAR ELECTRIC PROPULSION

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Abstract

Advanced solar electric power systems and electric propulsion technology constitute viable elements for conducting human Mars transfer missions that are roughly comparable in performance to similar missions utilizing alternative high thrust systems, with the one exception being their inability to achieve short Earth-Mars trip times. A modest solar electric propulsion human Mars scenario is presented that features the use of conjunction class trajectories in concert with pre-emplacement of surface assets that can be used in a series of visits to Mars. Major elements of the Mars solar electric transfer vehicle can be direct derivatives of present state-of-the-art Solar array and electric thruster systems. During the study, several elements affecting system performance were evaluated, including varying Earth orbit altitude for departure, recapturing the transfer stage at Earth for reuse, varying power system mass-to-power ratio, and assessing solar array degradation on performance induced by Van Allen belt passage. Comparisons are made to chemical propulsion and nuclear thermal propulsion Mars vehicles carrying similar payloads.

Introduction

When solar-electric propulsion (SEP) and power generation systems are matched to appropriate payloads for transporting and sustaining a crew on Mars, a mission architecture can be constructed that avoids some of the inherent challenges characteristic of high thrust systems (such as Mars aerocapture and cryogenic propellant management). This study proposes a SEP Mars architecture, specifying the payload elements necessary for scientific investigation of Mars and the crew systems (surface and transfer habitats) necessary for three-year missions.

One objective of this analysis is to consider the evolution of commercial SEP technologies in a way that would allow for some degree of commonality with a human Mars program. A modest Mission scenario is presented that features the use of conjunction class trajectories in concert with preemplacement of surface assets that can be used in a series of visits to Mars. With each mission opportunity an accumulation and coupling of the delivered surface assets (that support the crew and its science mission objectives) would lead to a growing Mars base from which more demanding surface science would be conducted by subsequent missions. Figure 1 shows a plausible Mars mission sequence and base build up schedule, with vehicles departing about every two years. Mars base build up would thus occur over several opportunities.

SEP Key Technology Elements

Several key elements of a SEP Mars transfer system serve to distinguish it from its high thrust rivals for interplanetary transfer. These elements or capabilities would include its megawatt power system, its large array size, and its requirement for long life operation. Technology issues for SEP Mars Vehicles would include the following:

- 1. High efficiency Solar cells
- 2. Low specific mass Solar arrays
- 3. Cell and array manufacturing technology
- 4. Array installation and deployment
- 5. Large scale efficient, long life Hall thrusters
- 6. Solar array pointing
- 7. Thruster cluster gimbaling
- 8. Life sciences for long trip time
- 9. Radiation degradation mitigation

Space Solar power generation technology is frequently characterized by its mass-to-power ratio (Alpha). For the purposes of this study, the Alpha value was baselined to 8.4 kg / kWe. The nominal electric propulsion performance is baselined as current Hall thruster operating at 2000 seconds specific impulse (Isp). Thruster specific mass is baselined at 3.4 kg / kWe. An advanced Isp value of 4000 seconds was also used in calculations to determine sensitivities to Isp.

Human Mars Mission and Vehicle Set

One objective of this task was to identify and evaluate a viable Mars vehicle fleet (set of Piloted and Cargo vehicles) that would have as its goal the emplacement of a modest surface base infrastructure. This infrastructure includes a 600 day crew surface habitat with its support systems, including ground power, science equipment and surface mobility rovers. The scenario chosen emphasizes cargo pre-emplacement; this approach reduces the required payload mass that must be taken on the piloted transfer stage. This is in part a concession to the nature of low thrust interplanetary transfers: by reducing the total vehicle mass for the piloted vehicles at Earth departure, spiral out and Earth-Mars heliocentric trip times can be minimized. It is less important to minimize cargo vehicle travel times. The payload elements are appropriately separated into three elements, and each is sent on its own transfer stage. The two cargo vehicles are sent on one-way only transfers; the piloted Mars transfer vehicle (MTV) is round trip capable and potentially reusable. The two cargo vehicles have roughly the same payload and are thus identical. For the architecture outlined herein, three transfer stages are used for each opportunity. Each Earth-Mars opportunity cycle uses vehicles identical in configuration, with the exception that later missions incorporate advanced components as they become available, such as more efficient arrays, higher Isp thrusters, and lighter structures.

First Cargo Mission Transfer Stage (Cargo 1)

The first element of the system, an unmanned cargo MTV, travels to Mars and propulsively captures into orbit. Its payload, a cargo lander, descends autonomously to the surface. This first mission delivers the crew surface habitation system. It is a descent stage carrying surface equipment, consumables, rovers, and a surface habitat as payload (Fig. 2). The lander mass is 58 metric tons (mt). The useful payload landed is 33 mt. Of this, the surface hab is 20.5 mt and the supporting surface systems and science equipment is 12.5 mt, including a pressurized rover.

The lander uses low energy aerobraking for descent; all but about 700 m/s is taken out aerodynamically, the remainder with rocket propulsion. Once landed, the surface habitat's environmental control and life support (ECLS) systems are autonomously activated and monitored for a period to validate proper operation. Improperly functioning elements are identified, and the system control software switches operation of malfunctioning components to backup units in the redundancy path of the ECLS system architecture. After the monitoring

period, the surface habitat enters a quiescent state until crew arrival. Replacement ECLS system elements are either located on board the surface habitat or are carried onboard the piloted vehicle, to be brought over for change out.

Alternatively, this first Cargo mission lander could remain in Mars orbit until the crew arrives. In that option the piloted MTV would rendezvous and dock with it in orbit, allowing the crew to board the habitat, and check the surface ECLS systems before descent.

The Cargo 1 lander illustration (Fig. 2) shows a cylindrical habitat mounted on top of a landing leg structure that cradles a rover. The descent engines are side mounted to create an open area just below the central portion of the surface habitat. This cradle configuration allows for immediate off loading of the rover without requiring any dedicated off-loading crane. The surface habitat lander carries additional surface mission hardware, surface power system hardware, and crew consumables such that its mass is equal to the total mass of the Cargo 2 lander. This approach allows the two unmanned Cargo transfer stages to be identical.

Second Cargo Mission Transfer Stage (Cargo 2)

The Cargo 1 mission is followed by another unmanned cargo stage, Cargo 2, carrying a 58 mt Mars ascent / descent stage. This vehicle captures into a prescribed Mars parking orbit and there awaits the arrival of the piloted MTV. Its ascent stage weighs 35.3 mt, carries a 4.1 mt ascent cab as payload and has propellant sufficient to ascent to a 250 km by 1 Sol period elliptical parking orbit. Ascent to orbit dV is 5,300 m/s. The descent stage weighs 13.3 mt, and its descent aerobrake weighs 9.1 mt (Fig. 3).

Both lander systems utilize low energy aerobraking on descent and both utilize current RL-10 O2 / H2 cryogenic propulsion technology (466 second Isp) for the final 700 m/s landing dV for propulsive deceleration, hover, translation, and touchdown. The ascent stage uses similar engines. Cryogenic O2 and H2 boil-off is maintained near zero via active propellant refrigeration assisted by surface power systems.

Piloted Transfer Stage

Having a crew transfer habitat as its primary payload, the piloted vehicle is configured for rendezvous with either or both of the cargo vehicles waiting in Mars orbit and return to Earth after the surface mission is complete. The large 40 mt crew transfer habitat contains consumables, outbound and inbound science equipment, a solar proton and galactic cosmic radiation shelter, avionics, communication, and data

handling systems sufficient for interplanetary navigation. The outbound and inbound payload mass for the Piloted MTV is the sum of the transfer habitat (40 mt) and the small Apollo style Earth Crew Return Vehicle (ECRV) mass (5.8 mt). Upon arrival and propulsive capture into a Mars parking orbit, this piloted vehicle is to rendezvous with the Cargo 2 vehicle. The crew then transfers to the ascent / descent lander and prepares it for descent. Once on the surface, the crew transfers to the surface habitat, activates its systems, and begins change out of faulty ELCS elements where necessary. After the checkout is complete, a portion of the crew returns to the ascent stage, placing it in a quiescent mode. It will await use at the end of the surface mission. After the surface mission the crew boards the ascent vehicle, ascends to orbit, and returns to the MTV for the inbound transfer to Earth.

After transfer to the transfer stage habitat, the crew separates and jettisons the propulsion stage of the spent ascent craft, leaving the ascent crew cab attached to the habitat. The crew cab serves as a backup control and communication center for the transfer stage, providing a modest supplemental habitable volume for the crew. After departing Mars orbit, the crew examines and catalogs surface samples and experimental data and readies the ECRV for Earth reentry during the inbound transit to Earth. As mentioned earlier, this Mars architecture uses the same three-vehicle set for each mission opportunity. Each MTV captures propulsively into the same Mars parking orbit 250 km by 1 Sol orbit, which has a Mars apoapsis of 33,000 km.

Effectiveness of SEP for Mars Missions

During the study, several elements affecting SEP system performance were evaluated, including the following:

- Varying Earth orbit altitude for departure
- Recapturing the MTV at Earth orbit for reuse
- Varying Isp
- Varying power system Alpha
- Including Van Allen Belt passage degradation on the Solar arrays
- Varying initial vehicle thrust-to-weight (T/W) at departure
- Varying power system level
- Varying Earth spiral out trip time

As a means of comparing the effectiveness of SEP technologies for human Mars missions, chemical propulsion and nuclear thermal propulsion (NTP) vehicles were also evaluated. Each vehicle type was configured to deliver the same payloads. Of the

various mission opportunities evaluated, only the year 2018 mission data will be presented in this paper.

Earth-Mars Trajectories

A 2018 departure, low thrust Earth-Mars Heliocentric trajectory characterized by a 255 day heliocentric outbound trip time, 600 day surface stay time, and a 156 day inbound trip time was used for the SEP vehicle analysis. Total heliocentric dV for this mission is 15,430 m/s. Earth spiral out time is dependant on several factors, primarily vehicle T/W and start orbit altitude. These will be discussed in more detail later. For reference, a generic transfer stage thrust-to-weight vs dV relationship is shown in Fig. 4 for one-way LEO-to-GEO transfers.

The comparable high-thrust 2018 mission trajectory chosen had a 233 day outbound time, a 600 day stay and a 158 day return trip time. Total heliocentric dV was 10,014 m/s. This value includes the Mars Orbit Capture (MOC) dV, which for the NTP vehicle was done propulsively. The chemical vehicle used aerocapture for MOC. Earth escape and Mars interplanetary dV's are listed in Fig. 5 for both low and high thrust trajectories for the 2018 opportunity. Because the high thrust trajectory chosen is not a fast trip time transfer (< 180 days outbound), arrival velocities at Mars are moderate. Figure 6 shows typical low thrust Earth-Mars heliocentric transfer dV as a function of trip time. Figure 7 shows low-thrust data for several different departure opportunities.

Baseline SEP Piloted Transfer Vehicle

Figure 8 contains weight data and a sketch of the baseline piloted SEP MTV. Component masses for both reusable and expendable vehicles are given. The propellant required for the reusable vehicle is about twice that of the expendable. This is attributable to the end-of-mission Earth spiral down propellant (42 mt), which is carried through the entire dV. The piloted vehicle is illustrated with the crew habitat module forward and the Xenon or Krypton tankage and thruster system aft. The arrays shown are intended to be representative only, and are not shown to scale. Their surface area is a function of cell efficiency, required power at Earth departure, and degradation effects of Van Allen Belt passage.

Typical array areas for the piloted vehicles under study are on the order of 4000 - 6000 meters squared (m2). Proportion by weight of the vehicles major systems is shown in Fig. 9 for both expendable and reusable vehicles. Figure 10 lists trip times and the various orbit parameters for the departure and capture points. For the baseline SEP, Earth spiral-out time (267 days) is comparable in duration to the Earth-Mars transfer time (255 days). End of mission

Earth spiral-down time for the reusable vehicle is 74 days.

SEP Cargo Transfer Vehicles

SEP Cargo vehicle propellant loads (Fig. 11) were calculated with these vehicles flying identical 2018 heliocentric trajectories as their piloted counterparts. Earth spiral times are the same because the thrust-to-mass ratios are identical. The expendable Cargo craft remains in Mars parking orbit; the reusable version, carrying no inbound payload, returns to its original departure orbit and awaits reuse at the next opportunity.

Earth Departure Altitude

The nominal SEP Earth departure point is 300 km circular LEO. Vehicles are delivered to LEO, assembled, and spiral out to Earth escape velocity before entering the heliocentric trajectory. Spiral-out times are on the order of 100's of days, so a significant duration is spent in Van Allen belt passage. For this reason the transfer code was set to apply a nominal power degradation value of 20% for all systems departing LEO that had spiral times of greater than 100 days. The VARITOP low thrust trajectory code was used to determine SEP trip times, dV, and required vehicle thrust-to-weight values at Earth Escape. Runs were also made at 40% and 0% degradation to determine sensitivities to vehicle mass. As a means of eliminating the array radiation degradation penalty, as well as to shorten the spiral out time, an alternative staging altitude was evaluated in which the SEP vehicle departs out of a 15,000 km circular orbit, above the radiation belt (Fig. 12). This altitude roughly corresponds to what is referred to as a Medium Earth orbit (MEO), so in the remainder of this section "MEO" and 15000 km circular orbit are used interchangeably. The low thrust dV required to depart Earth (departure altitude to escape velocity) from this higher altitude is roughly one half that required departing LEO (3,578 vs 7,185 m/s), and no array degradation is involved. For vehicles to be assembled in 15,000 km MEO, arrays may be deployed and all systems (including propulsion) might first be checked out and verified in LEO. The SEP vehicles may then be boosted to their departure altitude by high thrust boost vehicles. Solar arrays can be either drawn back in or left deployed for the boost to 15,000 km. Van Allen belt passage durations would be on the order of hours rather than months and degradation damage to the arrays would be insignificant.

An alternative to LEO assembly would consist of delivery of the vehicle elements by ETO launch vehicles to LEO, each of which would then be

boosted separately to the departure orbit before assembly. Joining of elements, array deployment, and vehicle systems checkout take place at the higher orbit before departure. Below is a more complete list of the advantages and disadvantage of departing from MEO:

Advantages of MEO Departure

- 1. SEP arrays are not deployed until after arriving at MEO orbit. This is above the radiation belt and thus belt passage degradation losses need not be applied
- 2. MEO departure eliminates the danger of assembly orbit decay due to atmospheric drag, important to large area, low ballistic coefficient configurations. Array areas were on the order of 6000 m2 for some vehicles evaluated.
- 3. The debris population at MEO is less than at LEO.
- 4. The local g-level at 15,000 km altitude is 0.871 m/s2, an order of magnitude less than at LEO (8.80 m/s2). Required thrust levels are much less at MEO altitude.
- 5. Earth spiral out time is reduced compared to LEO departure. Vehicle return back to its Earth departure altitude for reuse entails less dV to spiral down. For reusable SEP vehicles departing and returning from MEO, the total dV advantage compared to the LEO-to-LEO mode is on the order of 7,000 m/s. The power requirement for the baseline LEO departure SEP piloted craft is 2.9 MWe; for the MEO option, the requirement is for 1.9 MWe. Spiral out time is 267 days in comparison to 119 days, while heliocentric outbound Earth-Mars trip time is the same at 255 days. This occurs because both were configured to operate at a Thrust-to-Mass level of 0.00039 N/kg at the "Earth escape condition" which represents the beginning of the heliocentric transfer.

Disadvantage of MEO Departure

1. A high thrust boost stage is needed to get the SEP vehicle to MEO. The propellant load that would be needed to boost the SEP vehicle to MEO by a lower Isp high thrust stage would be greater than that required for the LEO to MEO krypton propellant if the SEP departed LEO.

The results of the comparison of LEO and MEO as staging points are summarized in Figs. 13 and 14. The analysis of a high-thrust vehicle to transfer the Mars vehicles to MEO was not completed as part of this study, and no final conclusions can be made.

Evaluation of Vehicle Reuse

The option of returning the SEP vehicle to Earth for subsequent reuse was also evaluated. Returning to MEO rather that LEO reduces propellant loading substantially, reduces the spiral down trip time and eliminates Van Allen belt passage degradation on the way down. For the nominal expendable mission, the Mars transfer stage flies an inbound trajectory that matches Earth Heliocentric velocity but does not spiral in. At this point, the crew boards the ECRV and returns via aerodynamic reentry to splashdown or LEO. In this mode no spiral down propellant is carried and the vehicle is expended after one use. However, because of the high Isp, SEP vehicles are good candidates for reusability. For the case of a piloted MTV that is returned to Earth orbit, refurbishment for reuse would consist of re-supplying propellant and providing crew consumables. The expensive solar array power generation system, electric thrusters, and the large crew transfer habitat are all reusable systems. Reusable SEP vehicles may have an economic advantage over their expendable counterparts because of the potential cost savings enabled by multiple vehicle reuse. Because of the lower Isp of the Chemical and NTP concepts, return to Earth orbit (LEO or MEO) is much more expensive in terms of propellant. SEP Initial Mass in Low Earth Orbit (IMLEO) is plotted for LEO departures and LEO returns in Figs. 15 and 16, and for MEO departures and returns in Fig. 17. Both the latter figures indicate mass variation with Alpha.

Reuse Discussion

For the LEO departure / LEO return mode a total of 388 mt of propellant is required for the 2018 piloted SEP mission; of this amount, 43 mt is required for the spiral down to LEO. As expected, a similarly configured MEO to MEO vehicle not counting the required boost to MEO stage, is significantly lighter; the total propellant load is less than one-half that of the LEO vehicle (total propellant is188 mt); of this only 15 mt is needed for spiral down to 15,000 km. See Table 1 for more information.

The propellant required of a lower Isp chemical upper stage to boost the SEP vehicle to the 15,000 km orbit would eliminate the mass advantage. The cost savings associated with reuse of the SEP elements must be traded against the expense of these boost stages. If there already exists a large high thrust upper stage (for LEO-GEO, or LEO-Moon) that could be used for this purpose, then the expense for its development could be avoided. Such a stage was not defined for this study. It is recommended that a MEO mission OTV system be defined and possible synergism with other missions be identified.

Resupply Requirements

With reuse, the production (over the course of a multiyear Mars program) of the number of large arrays, thrust modules and transfer habitats would be reduced. There would also be a saving in the launch of hardware. Figure 1 depicted a plausible Mars base build up scenario, with vehicles departing about every two years. Base build-up would thus occur over several opportunities, and a fleet of three SEP vehicles could be reused for a number of opportunities. Table 1 indicates the propellant resupply requirements for both LEO and MEO departures. After recapture, a followon mission with the same vehicle can be done with a resupply mass of less than one-half the original vehicle IMLEO. The expensive power system, structure and transfer hab can all be reused. For example, a 274 mt MEO to MEO reusable piloted MTV could fly the next opportunity with the addition of 143 mt of propellant, in an expendable mode, or with 188 mt of propellant, with return and recapture into MEO with the potential for yet another reuse.

Power System Efficiency

Alpha ratio was varied to determine its influence on IMLEO. Figures 16 and 17 illustrate the relationship, and it can be seen that variation of Alpha has a marked effect on the mass of the reusable vehicles departing the LEO altitude. For example, IMLEO grows by 150 mt for an Alpha increase from 8.4 to 12 kg / kWe for the 2018 reusable piloted MTV. Its expendable counterpart grows by less than 40 mt. The heavier vehicles departing LEO (8.7 m/s2 local gravity) showed more sensitivity to power system growth than those departing MEO, where local gravity (0.86 m/s2) is lower by an order of magnitude. There, the thrust required is much reduced for a given vehicle T/W, and the corresponding power system mass is much lower. For one case evaluated, the power system mass at 15,000 km was only 27 percent of a similar vehicle departing LEO.

If power system mass is a driver of total system cost on the same order as it is for IMLEO (a traditional yardstick for estimating cost), then this benefit (when coupled with the two other benefits of MEO departure – no array degradation and shorter spiral out times), make the MEO mode worthy of consideration. Any detailed analysis, however, must factor the cost of procurement and operation of the high thrust boost to MEO vehicles.

Power Level and Spiral-Out Time

Spiral-out time as a function of power level is given in Figs. 18 and 19. The baseline reusable MTV uses a 2.91 MWe system that requires 267 days to spiral out.

Doubling the power level would only reduce the spiral out time by 30 percent.

Effect of Array Degradation

Figures 20 and 21 illustrate SEP mass and area variation with array degradation from multiple passes through the radiation belt. Based on Fig. 20, every 1% of array degradation would necessitate a 2.5 mt increase in IMLEO. From Fig. 21, for every percentage of degradation, about 75 m2 of area must be added to the array system to make up for loss power as compared to an ideal, completely radiation resistant array.

Isp Sensitivity

Hall thrusters at 2000 sec Isp and 44 percent efficiency were baselined. Isp was increased to 4,000 sec to examine sensitivities to vehicle IMLEO, power system mass, and trip time. Because increases in electrical power are necessary for higher Isp, raising Isp has the contradictory effect of lowering the propellant requirement while increasing the power system mass for a given thrust-to-weight. Cargo vehicle IMLEO vs Isp data given in Fig. 22 illustrates the tradeoff involved in operating at higher Isp. The difference is significant for the LEO departure vehicle. The propellant savings due to increasing Isp to 4000 sec is roughly equivalent to that obtained by raising the departure altitude to MEO.

An increase to 4,000 sec does not always results in lower IMLEO; at least one vehicle investigated required a higher IMLEO with the increase to 4000 sec Isp. When the total thrust required at departure is relatively large, the increase in power system mass can offset the propellant savings.

Thrust Level

Thrust level for the low thrust systems is set by considering both the requirement for a minimum vehicle thrust-to-mass ratio at Earth escape in order to achieve the prescribed Earth-Mars trajectory, and the initial thrust-to-weight ratio at the Earth orbit departure point. As an example, for the 2018 heliocentric trajectory chosen for this analysis, a thrust-to-mass ratio of at least 0.000386 N / kg is required at the beginning of heliocentric flight. Additionally, it is necessary that the Thrust-to-weight at Earth departure be at or above 0.00001. Given these two general limitations, a minimum required thrust level can be determined. The largest thrust level prescribed by these two requirements is the level needed. A given thrust level implies a thruster jet power, and given a thruster efficiency, the electrical power required into the thrusters can be determined. The power system weight and array size is determined

from this factor and several others, including the Alpha of the power system, as well as the efficiency of the arrays. The expected power loss due to array degradation over the duration of the mission is also a consideration.

Power Beaming to the Mars Surface Base

An option exists for the solar arrays of SEP transfer vehicles parked in Mars orbit to track the sun and beam down power to a receiving antenna on the surface, where it can be converted to electrical power. The MTV's parking orbit can be set so as to be in a 1 Sol period circular orbit (Mars synchronous orbit), positioned directly overhead the landing site at all times, allowing for the option of continuous power beaming to the surface.

Comparisons to Chemical and Nuclear Propulsion

Comparisons were made of SEP vehicles to Chemical propulsion and NTP vehicles carrying identical payloads. The chemical piloted MTV consists of three stages. Each use LO2 / LH2 RL-10 propulsion technology operating at 466 seconds Isp. A core stage does the Mars Orbit capture (MOC) and TEI burns. This stage has the crew transfer habitat as its payload. This stage is boosted to trans Mars injection (TMI) velocity by two nearly identical Earth departure stages. The core MOC/TEI stage carries adequate propellant reserves to make up for hydrogen boiloff over the duration of the mission. Because of the modest Mars arrival velocity, high energy aerocapture into orbit is not used; the vehicle propulsively captures into a 250 km by 1 Sol orbit.

The NTP vehicle consists of one stage, and uses the same three 15,000 lbf thrust nuclear engines for all major burns. The engine cluster is located at the aft of the vehicle, and the habitat at the front, with the two separated by a long truss structure, from which are attached separate TMI, MOC and TEI propellant tanks. The TMI and MOC tanks are jettisoned after use to reduce mass for subsequent burns. NTP Isp is 925 seconds, and the total burn time required of the engines for one Mars mission is on the order of 3 hours. The nuclear engines are each rated for an estimated 10 hours of life, so they offer the potential for reusability. Figures 23–26 set forth IMLEO comparisons for the three vehicle types. SEP IMLEO's are generally comparable to the all chemical systems (Fig. 23) when recapture at Earth is not a consideration; when it is, SEP vehicles have the advantage of lower total mass.

The NTP piloted vehicle requires significantly less IMLEO than either the all chemical or SEP vehicle (Fig. 24 and 25). However, at an Isp of 4000 sec the SEP vehicle is comparable for the

expendable mission (right most bar in Fig. 25) and significantly better (less IMLEO) for the reusable mission (Fig. 24 right bar). In all cases the SEP requires more inert mass. The benefits of SEP systems seem to be more favorable, in comparison to the high thrust alternatives, in a reuse mode, because of its higher proportion by weight of inert mass. This factor leads to consideration of the cumulative effects of multiple pass array degradation, and means of mitigating such in order to extend the useable life of the power system. Enabling a single SEP vehicle multiple mission capability would allow it to overcome its IMLEO disadvantage to NTP technology (as it has been configured in this analysis). It has already been shown that at 4000 sec Isp, the SEP is comparable to NTP. A 2018 piloted NTP vehicle sketch and mass statement is given in Fig. 27.

Other Factors Considered in Assessment

Factors other than IMLEO must be considered, however; the NTP has one distinct advantage in that it requires no spiral-out time and can fly fast trip time missions. The SEP advantage involves its power system technology, which has a broader applicability to other programs and missions. Because the NTP technology has a narrower field of applicability, it

seems unlikely that NTP would be developed solely for Mars transfer propulsion if commercially derived space solar power systems were available. A synergistic approach to generating a solar power system technology program path that considers concurrently the needs for both interplanetary transfers and space based power systems seems advisable. Determining an overall architecture preference for human Mars missions is a complex endeavor, with many more issues to be considered than have been covered in this assessment. See Refs. 1 and 2 for more in-depth Mars architecture studies.

References

- 1. Donahue, B., and Cupples, M., "Comparative Analysis of Current NASA Human Mars Mission Architectures," AIAA Paper 2000-3215, AIAA Joint Propulsion Conference, Huntsville, AL, July 2000.
- 2. Donahue, B., "Human Mars Missions: Cost Driven Architecture Assessments," AIAA 98-5285, Defense & Civil Space Programs Conference, Huntsville, AL, October 28, 1998

Table 1. Resupply Mass Required for 2018 SEP MTV Follow-on Missions

Departure Altitude	Return Altitude	Propellant	Transfer Hab + ECRV	Power Sys + Dry Mass	Follow on (2 nd) Mission Type
LEO	Expended Veh	184.7	45.8	37.5	n/a
LEO	to LEO Veh	388.7	45.8	66.8	n/a
	- Resupply Required	388.7	reused	reused	Return again LEO
		235.3	reused	reused	Expended
MEO	Expended Veh	131.1	45.8	30.8	n/a
MEO	to MEO Veh	188.7	45.8	39.5	n/a
	- Resupply Required	188.7 143.0	reused reused	reused reused	Return again MEO Expended

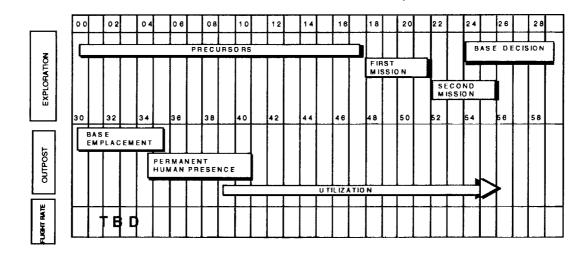


Figure 1. Candidate Human Mars Missions Sequence.

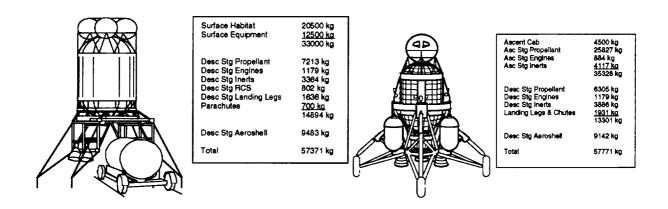


Figure 2. Mars Cargo 1 Lander

Figure 3. Mars Cargo 2 Piloted Asc / Desc Stage

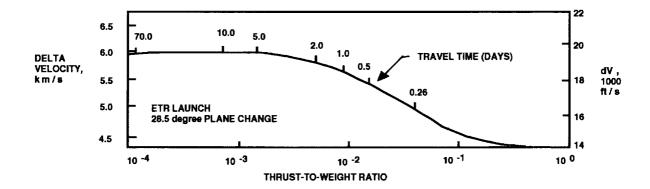


Figure 4. Typical LEO to GEO dV as a function of Initial Thrust-to-Weight.

EARTH DEPARTURES	High T	hrust dV	Low Thrust dV		
LEO -Earth Escape		3124	7185		
LEO -15.000 km circ		3150		'a	
15,000 km -E Escape	1789		3578		
HELIOCENTRIC TRIP TIME	360 days	391 days	350 days	411 days	
LEO -Trans Mars Injection	4200_	3635		_	
Mars Orbit Insertion	1159	1100			
Earth Escape-Mars Capture			11804	7716	
M Spiral to/from 250km x 1Sol			446	446	
Mars Escape-Earth Capture			11539	7716	
Trans Earth Injection	1314	1314			
Earth Orbit Insertion	3800	3800			
E Spiral Down to LEO			71	85	
E Sprial Down to 15,000 km circ			35	78	

Figure 5. Delta Velocity set for the Selected 2018 Mars Trajectories Utilized in Analysis

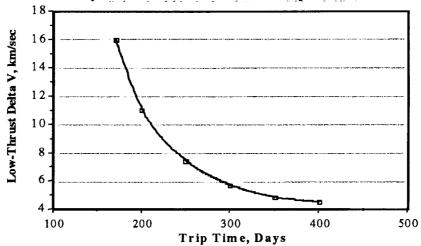


Figure 6. Typical Earth-Mars Low Thrust Delta-V vs Trip Time

Earth-Mars Helio. Departure Date	Helio. Delta-V km/s	Helio. Flight days	Mars Arrivel Date	Mars-Earth Departure Date	Helio. Delta-V km/s	Flight time days	Earth Arrivei Date
12 Jan 2014	14.850	209.5	10 Aug 2014	15 Jan 2016	12.375	209.9	12 Aug 2016
05 Mar 2916	13.552	189.0	10 Sep 2016	11 Apr 2019	11.205	177.2	05 Oct 2018
13 May 2018	11.804	169.2	29 Oct 2018	17 Jun 2020	11.539	181.1	15 Dec 2020
09 Apr 2018	7.716	255.8	21 Dec 2018	17 Jun 2020	7.716	273.9	07 Feb 2021
17 Jul 2020	12.165	180.7	14 Jan 2021	22 Jul 2022	12.740	209.5	16 Feb 2023
03 Sep 2022	13.777	208.4	30 May 2023	14 Aug 2024	13.360	233.3	04 Apr 2025
11 Oct 2024	14.852	223.5	23 May 2024	17 Sep 2026	14.243	235.0	10 May 2027
17 Nov 2026	15.297	224.9	30 Jun 2027	24 Oct 2028	13.928	233.9	15 Jun 2029
26 Dec 2028	15.118	215.6	30 Jul 2029	16 Dec 2030	12.736	222.9	27 Jul 2031
12 Feb 2031	14.170	197.7	28 Aug 2031	22 Feb 2033	10.356	210.1	20 Sep 2033
14 Apr 2033	12.416	175.4	07 Oct 2033	06 May 2035	9.155	208.7	01 Dec 2035

Figure 7. VARITOP Code Heliocentric Trip Times, dV and Departure Dates

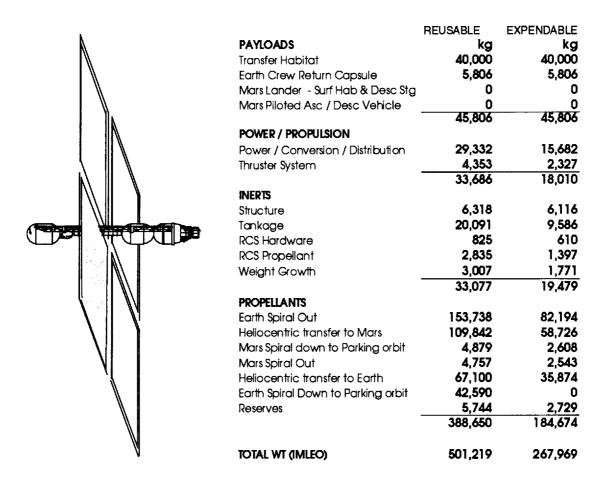


Figure 8. 2018 SEP Piloted Vehicle Weight Statement (arrays not shown to scale)

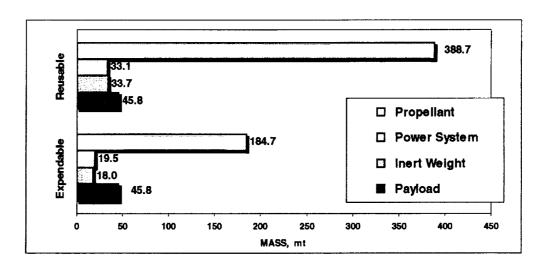


Figure 9. Mass Break Out by Major Subsystems, 2018 SEP Piloted MTV

Phase	initial Orbit (km)	Final Orbit (km)	Initial (mt)	Mass Propel (mt)	Final (mt)	Time (days)	Tot Mission Duration (days)	Detta-V (km/s)
LEO to Earth Escape Earth Escape to Mars Orbit Capture Mars Surface Stay Mars Capture Orbit to Matching Earth Vel Matching Earth Arrival Vel to Earth LEO	300 x 300 E Escape — 250 x 33,000 Mars E Escape	E Escape 250 x 33,000 Mars — E Escape 300 x 300	501 347 232 232 160	153 114 0 71 42	347 232 232 160 118	267 255 600 156 74	267 522 1122 1278 1352	7.15 8.16 0 8.16 7.15

Figure 10. 2018 piloted Mission Parameters

CARGO 1 & 2 Vehicles 2018 Mission		ENDABLE led at Mars
SPIRAL OUT TIME	267 days	267 days
TRIP TIME HELIOC	366 days	255 days
E ESCAPE T / Mass	3.9 x 10-4	3.9 x10-4 N/kg
Array Power	1.70 MWe	1.07 MWe
Array Area	3711 m2	2346 m2
Total SEP Veh Mass	292.1 mt	184 .7 mt
Propellant Mass	194.3 mt	100.4 mt
Power / Propul Mass	19.6 mt	12.4 mt
Inert Mass	20.2 mt	13.9 mt
Payloads	58.0 mt	58.0 mt

Figure 11. Baseline 2018 Mars SEP Cargo Vehicle Weights

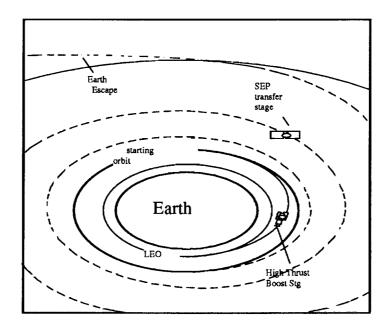


Figure 12. MEO Departure Mode Diagram

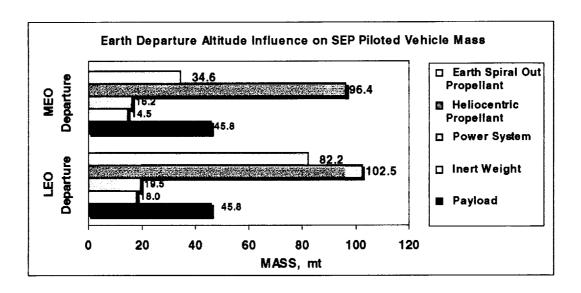


Figure 13. MEO and LEO Departure System Weights

2018 MISSION	DEPARTURE ORBIT ALTITUDE						
PILOTED MARS SEP	•	000 km Circ	LEO 300 km (Array Degradation 20%)				
VEHICLES	(Above Va	an Allen Belt)					
	Expended	Reused	Expended	Reused			
SPIRAL OUT TIME	119 days	119 days	267 days	267 days			
E DEPARTURE T/Wt	3.6 x 10-4	3.6 x 10-4	0.3 x 10-4	0.3 x 10-4			
TRIP TIME HELIOC	411 days	411 days	410 days	410 days			
Earth ESCAPE T / Mass	3.9 x 10-4	3,9 x 10-4	3.9 x 10-4	3.9 x 10-4			
Array Power	1.47 MWe	1.94 MWe	1.56 MWe	2.91 MWe			
Array Area	2,681 m2	3,538 m2	2,404 m2	6,368 m2			
Total SEP Veh Mass	207.7 mt	274.1 mt	268.0 mt	501.2 mt			
Earth Spiral Out Propel	34.6 mt	45.7 mt	82.2 mt	153.7 mt			
End of Mission Spiral In	0.0 mt	15.5 mt	0.0 mt	42.6 mt			
Heliocentric Propellant	96.5 mt	127.5 mt	102.5 mt	192.3 mt			
Power / Propul Mass	14.5 mt	19.2 mt*	18.0 mt	33.7 mt*			
Inert Mass	16.3 mt	20.4 mt	19.5 mt	33.1 mt			
Transfer Habitat & ECRV	45.8 mt	45.8 mt *	45.8 mt	45.8 mt*			
OTV Boost Stg	TBD	TBD	n/a	n/a			

^{*} Returned to Earth Departure Orbit for Reuse

Figure 14. MEO and LEO Departure Mode Comparison.

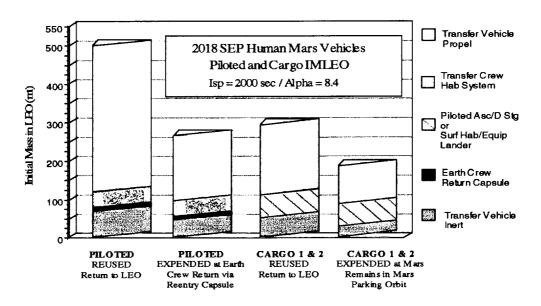


Figure 15. Weight Differences of Reusable and Expended Vehicles

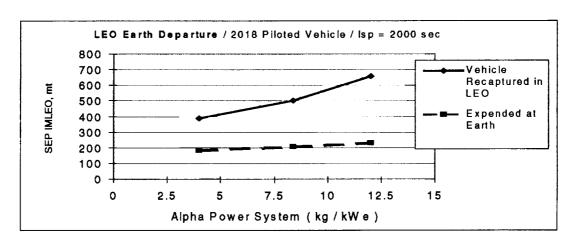


Figure 16. Power System Sensitivities: LEO Departures.

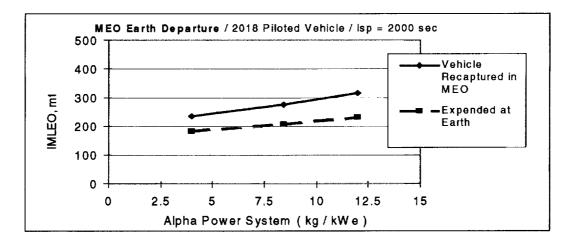


Figure 17. Power System Sensitivities: 15,000 km Circular Orbit Departures.

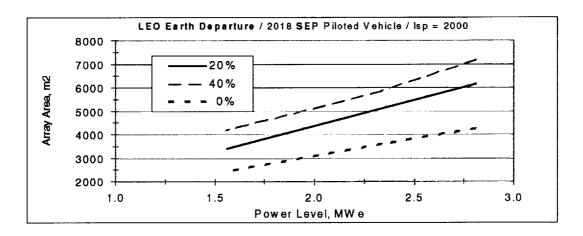


Figure 21. Array Size vs Power System and Degradation Percentage

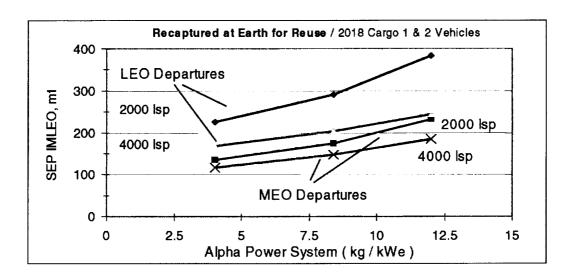


Figure 22. Mass Reductions Enabled by Doubling the Isp for Cargo Vehicles

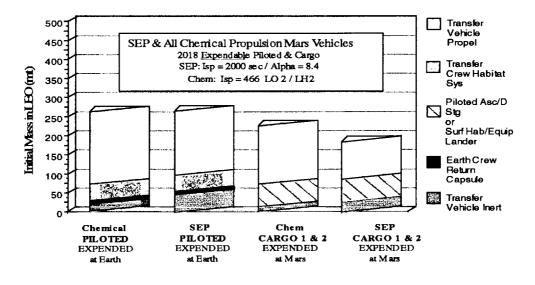


Figure 23. All Chemical and SEP Transfer Vehicles Comparison

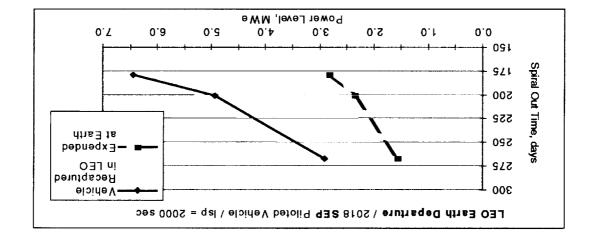


Figure 18. Spiral Out Trip Time as a Function of Power Level LEO Departures

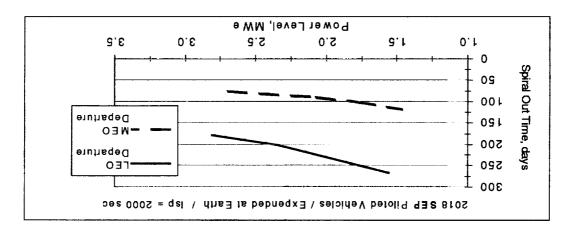


Figure 19. Spiral Out Time vs Power Level for Expendable Vehicles

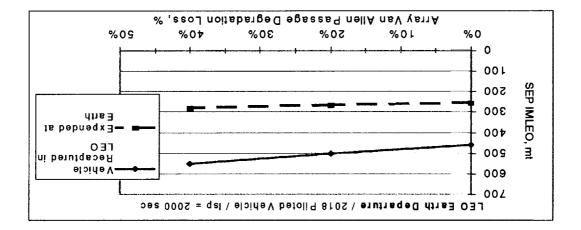


Figure 20. SEP Vehicle IMLEO vs Array Radiation Degradation

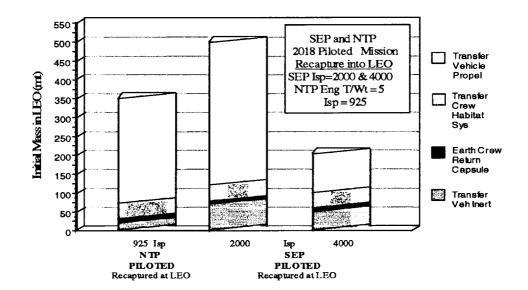


Figure 24. NTP, Chemical and SEP Data for Expendable Piloted Vehicles

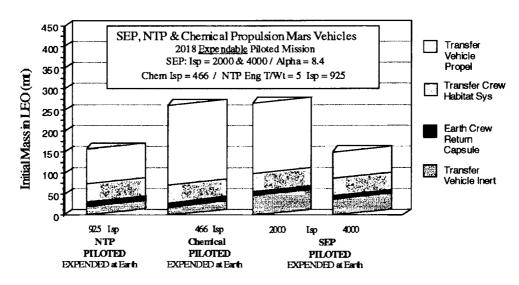


Figure 25. NTP and SEP Vehicle Comparison for Reusable Piloted Stages.

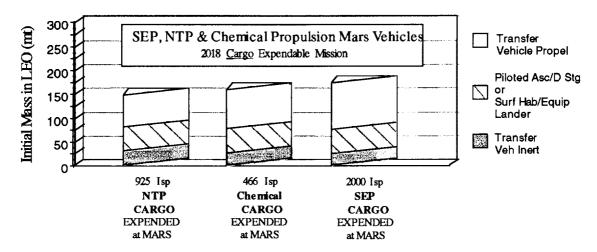


Figure 26. NTP, Chemical and SEP Comparisons for Expendable Cargo Vehicles

	STRUCTURES	kg	RCS		PROPELLANT	
	Adapter struct	450	Tankage	340	Bolloff	6,802
	Interstage / Truss	13,245	Thursters	1,645	TMI usable prop	95,374
	1 Thrust Struc	1,037	Pressurization	430	MOI usable prop	46,235
	Miscellaneous	222	Propeliant	2,277	TEI usable prop	23,941
T	-	14,954	•	4,692	EOI usable prop	52,975
L A	TANKAGE		WEIGHT GROWTH		· · · · · · · · · · · · · · · · · · ·	225,327
	Fuel tankage	27,772	On dry weight	4,512		
(♣) }	Fuel insulation	4,306	STG DRY WT	70,824	STG GROSS WT	309,262
18	N2 pressurization	224				
	<u>-</u>	32,304	RESERVES		TOT P/LS WT	45,806
	PROPULSION		Residuais	1,093		
(() 	Engine: 3	9,644	Reserves	3,278		
(🖺)	Thrust vector contro	482	Cool down	8,741		
	Propulsion related,	290		83,935	•	
展	Radiation shields	3,000			TOT VEH WT	355,068
	-	13,416				

Figure 27. 2018 Nuclear Thermal Propulsion Reusable Piloted Mars Vehicle